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SPACETIME—A MIDEX PROPOSAL TO TEST EINSTEIN'S EQUIVALENCE PRINCIPLE

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ABSTRACT

SpaceTime is a MIDEX-class proposal. This paper describes the mission proposed for MIDEX 98. SpaceTime uses a Jupiter gravity-assist trajectory to send a spacecraft past the sun at approximately 4 solar radii. This allows for the search of a violation of Einstein's Equivalence Principle by studying the differential redshift of 3 atomic clocks. This low-cost mission is made possible by the new application of technology (i.e., carbon-carbon heat shield) in conjunction with a simple spacecraft design utilizing a high degree of inheritance.

1. SCIENCE

1.1 Scientific Goals and Objectives

Do all neighboring clocks keep the same time, or do they show different rate shifts with location in space and time? Are nature's constants really constant, or do they change with time or location in the universe? How can gravity be reconciled with the quantum theory? What lies beyond the Standard Model which describes 3 of the 4 known forces?

These are some of the questions that probe the heart of our present-day understanding of the physical universe. Underlying all these questions is the role of gravity, the weakest fundamental force in nature, yet the most dominant force that, by virtue of its long-range interaction, determines the structure and evolution of the universe. Currently, the most successful theory of gravity is Einstein's General Relativity. Despite its success, General Relativity, formulated by Einstein as a metric theory which also describes the geometry of space and time, may break down if it is to be merged with the other forces and the most successful formalism in all of physics—the quantum theory. Most Grand Unification theories which include the quantum theory of gravity require modifications of, or additions to, General Relativity and its pure tensor-field description of gravitational interactions.

The scientific objective of this mission is to investigate the influence of gravity on clock rates—time—and to search for new physics with a simple instrument, and with a sensitivity beyond the current knowledge, to experimentally probe the limited validity of Einstein's Equivalence Principle. The Equivalence Principle is the grand hypothesis of Albert Einstein and is the cornerstone of every metric theory of gravity, including General Relativity. Simply stated, the Equivalence Principle asserts that the local gravity is exactly equivalent to an accelerated frame of reference.

There are several important implications of this hypothesis (which was referred to by Einstein as "the most profound realization of my life"), including the concept that the measured geometry of space and time is a manifestation of the gravitational field and the prediction that all clock rates are affected universally by any nearby matter. An observed deviation of this behavior is the sign for the limit of validity of metric gravity, and the clue that some other unknown force in nature exists, beyond the current Standard Model.

Precision experimental tests of the Equivalence Principle date back to Eotvos in 1922 (Figure 1-1). These tests probe the validity of the Equivalence Principle by diverse, yet generally believed equivalent, techniques. One of the most direct techniques is the comparison of clock rates to test the predicted universal redshift that all collocated clocks experience near a gravitating body (mass). This change in rate is quite small on the surface of the Earth, about 10^{-16}/meter of height, but can become progressively larger with the size of the nearby mass. This effect is essentially the same physical mechanism which leads to black hole event horizons, where clock rates approach zero as compared to other clocks farther away from the black hole. A precision measurement of the "gravitational redshift" of clocks has been previously performed by NASA's Gravity Probe-A experiment in 1976 (Figure 1-1). In that experiment, the frequency of a hydrogen maser clock launched into suborbital flight was compared with another hydrogen maser on the Earth surface [Vessot, et al., 1980]. This "absolute" measurement of the gravitational redshift required one-way and two-way radio links between the two clocks to achieve a precision of almost a part in 10^4.

In this mission, we will perform differential measurements of the clock redshift to a part in $10^{10}$of the shift experienced by three physically different ultrastable atomic clocks on a common spacecraft sent deep into the strongest gravitational potential available in the solar system—to within 4 solar radii of the Sun. With this configuration the frequencies of the clocks will be directly compared to each other onboard the spacecraft, eliminating the need for 1) high-precision Doppler links to a ground clock for clock-rate comparisons or orbit determination, and 2) consideration of residual errors due to limitations of modeling signal propagation through the interplanetary and atmospheric media. Any difference in the fractional frequency shifts of the clocks will indicate a breakdown of Einstein's Equivalence Principle and will signal the existence of a new interaction in physical law beyond the present day Standard Model of matter and fields.

At four solar radii, the Sun's gravitational potential (divided by c^2, where c is the speed of light) is 5×10^{-7}, so clocks accurate to about 10^{-16} will enable measurement of differential frequency shifts at the level of 10^{-10} of Einstein's universal prediction [Nordtvedt, 1996]. SpaceTime has a unique ultrastable "tri-clock" which consists of three independent atomic clocks operating in the same environment. Each of the clocks is based on a different atom [Prestage and Maleki, 1994]. This architecture allows for removal or cancellation of all errors due to common sources such as environmental perturbations. By operating all three clocks with the same local oscillator, the oscillator noise, an important source of degradation to high-stability atomic clocks, is also eliminated. Finally, the frequencies of the three different atoms in the tri-clock will be compared pairwise in a "three-cornered hat" configuration to determine the characteristic relative response of each atom to Equivalence Principle violating fields. This approach can lead to information about the nature of the detected field. In this manner, SpaceTime performs a test with an *unambiguous* outcome probing the validity of metric gravity

Another unique feature of the SpaceTime mission is that in its approach to the Sun, the spacecraft attains a speed of about 1/1000 of the speed of light. With this speed the SpaceTime spacecraft will be the fastest human-made object in the universe. Furthermore, during the 16 hours of solar passage the clocks' velocity with respect to the cosmic frame changes by a factor of almost a thousand more over the similar test performed by Turneaure et al (1983) which relied on the rotating Earth for velocity change. So, we test this possibility of velocity dependence with an increased sensitivity of almost 10^3, the ratio of the two velocities. We also gain over an order of magnitude more sensitivity (compared with Turneaure, et al) from the improved stability of our clocks.

1.2 Science Implementation

1.2.1. Instrumentation/Payload

In the strong gravitational potential at four solar radii, time runs slower than on Earth by about one half microsecond per second. Three atomic clocks based on hyperfine transitions of Hg^+ (Z=80), Cd^+ (Z=48), and Yb^+(Z=70) are different in their electromagnetic composition and will be simultaneously monitored during a solar flyby to determine whether clocks of different physical makeup will measure the same time interval in the strong spacetime curvature near the Sun. The atomic clock hardware for the SpaceTime mission is a modification of the linear ion trap frequency standard (LITS) currently being deployed in NASA's Deep Space Network stations worldwide. A laboratory prototype has shown ultrastable operation in a package far smaller than other clock technologies and represents the state of the art for atomic clocks.

The instrument for SpaceTime is composed of three ion trap clocks in a package where much of the hardware is common to all three. Because some of the clock systematic frequency perturbations will be common to all three clocks and will have a characteristic signature that can be identified and removed from the difference of the clock frequencies, relative stabilities to 10^{-16} in the intercomparison can be reached, making a 10^{-10} test of the Equivalence Principle. The local oscillator (LO) will simultaneously interrogate each of the three clock transitions, thereby removing LO noise in the intercomparison and greatly improving short-term clock noise so that 10^{-16}resolution in the difference in clock rates can be obtained within the 15-hour close encounter. Because ion trap-based clocks are relatively immune to temperature and magnetic field changes, a simple, robust electronics package is sufficient for ultrastable operation.

1.2.1.1 Clock Background

The modified mercury ion atomic clock developed at JPL is the primary instrument and payload for the mission described here. This ultrastable atomic clock is a modification of the very successful linear ion trap based frequency standard now being deployed at NASA's DSN stations worldwide. Altogether, ten mercury ion trap clock units have been built in the JPL Frequency Standards lab, all showing stability well into the 10^{-16}range. In the present application, we propose to investigate whether atomic clocks based on different atoms will keep the same rate in the strong gravitational environment near the Sun. The Equivalence Principle predicts that all such clocks will measure the passage of time at the same rate, that is, that gravitational time dilation or red shift will influence each of the three clocks of this experiment equally.

The experimental method used here is much simpler, more sensitive and most importantly, far less expensive than a traditional redshift experiment where one clock on Earth is compared to another clock in deep space at four solar radii. To carry out such an absolute redshift experiment to the 10^{-10} level, the solar flyby trajectory would need to be measured to 10^{-10}, i.e., to about 30 cm to accurately predict the expected redshift. Similarly, the velocity of the s/c, at 0.1% of the speed of light, would Doppler shift the link and require a 10^{-13} determination of the s/c velocity. The stable clock signal would have to be sent to an Earth station where the comparison to an equivalent clock would be made. All noise sources along the link signal to Earth, through the solar wind, ionosphere and troposphere would have to be less than 10^{-16}. This traditional redshift approach would be much more expensive and less sensitive than the SpaceTime in-situ measurement proposed here.

1.2.1.2 Mass Estimate

The space clock layout is very similar to the laboratory prototype which has shown excellent stability (5×10^{-16}) and has demonstrated the viability of the LITE architecture with mercury. The tri-clock physics package will occupy $10\times23\times44$ cm^3 and the electronics will occupy an equal volume and can be partitioned into two or more parts to accommodate s/c layout constraints. Most of these physics package items serve the same function as in a cesium clock and the electronics modules will be very similar to a cesium clock in GPS satellites. We have made a more complete comparison in a separate paper [Prestage and Maleki, 1994] where we estimate a total mass of about 11 kg for a single LITE clock based on Hg. Most of the material for the three-clock apparatus will serve all three clocks, e.g., magnetic shields, solenoid, lamp driver, etc. In this way, the 20-kg mass budget for the space clock payload can be met with adequate margin.

1.2.1.3 Power Estimate

Just as the mass for a space qualified LITE Hg^+ standard could be estimated from the analogous cesium space clock, so too can the power. Block IIR GPS cesium standards are the current state of the art for space clocks and have undergone several iterations of design refinement. These clocks consume less than 23 watts after warm-up and since most of the electronics is common we will use the 23 watts [Wisnia, 1992] as a baseline and add to that the additional electronics required for the LITE standard. The primary addition is the three radio frequency discharge lamps, which with careful design can be operated with a single resonator/exciter with less then 5 watts. Another addition is the radio frequency voltage for the trapping fields which can be done with less than 1 watt as in the space qualified mass spectrometer. This brings the total power consumption estimate to about 28 watts.

One of the many reductions of mass and power is by use of field-emission filaments as used in ion-propulsion systems. The space qualified heritage of these field emitters will insure a robust replacement for our resistive filaments currently in use.

These mass and power estimates have been reviewed and endorsed by the industrial partner for the LITE space clock electronics package, Frequency and Timing Systems, Inc. Their experience with space-based GPS cesium clocks will enable the development of a space-worthy triple clock within the constraints of this mission.

1.2.1.4 Magnetic Shielding

The suppression of systematic frequency pulling can be applied to variations of the solar magnetic field along the s/c trajectory. This approach will save mass and power in magnetic shielding. A set of two layers of magnetic shields will enclose the clock resonance tube as shown in the figure. An additional double layer will house the final package. Since the unshielded Hg^+ atom sensitivity is about 2×10^{-13}/mG (at an operating point of 50 mG), 20×10^{-13}/mG for Yb^+, and 15×10^{-13}/mG for Cd^+ a shielding factor of 10^7 is required to reduce a 1-gauss solar-field variation during the s/c flyby to below a part in 10^{16} relative clock stability. A 1-gauss field variation might be expected during the solar flyby as shown in the data from reference [Physics of the Inner Heliosphere]. This level of shielding is very difficult to achieve for the mass and power budget. The differential response of the three clocks to a common field variation has a characteristic signature that will identify this systematic shift and will enable its removal in post analysis.

A record of the ambient fields at the site of the clock, as supplied by a magnetometer, is used in post analysis to correct the clock frequency differences according to the pre-launch calibration data. Another method to measure magnetic field changes involves measuring field-sensitive atomic Zeeman transitions. This is a relatively simple and often used technique and requires no additional hardware.

1.2.1.5 Radiation Environment

The radiation environment at 4 solar radii is primarily electrons and protons from the solar wind. These charged particles are low energy; typically in the range of tens of electron volts with no substantial penetrating power. Furthermore, the Sun has no stored charged particle radiation belts as with the Earth and Jupiter. A special concern is radiation effects on the photomultiplier (PMT) tubes, used for measuring atomic fluorescence from the clock ions. Unwanted charged particle counting at high rates while photon-counting of atomic uv light fluorescence could diminish clock performance. However, the photomultiplier tubes of the LITE clock are located deep inside the instrument package with multilayers of shielding provided by the spacecraft structure, the clock housing and magnetic shielding, and the optical system framing. During the solar flyby, there will be no substantial degradation of clock performance from charged particle flux onto the photomultipliers. *The clocks will not be operating during the gravity assist at Jupiter so that radiation encountered there will cause no harmful effects.* The vacuum tube photomultipliers contain no solid-state electronics and achieve high-gain electron multiplication with multiple metallic electrodes. The photocathode is coated with low work function material and will not be damaged during the Jupiter radiation belt passage.

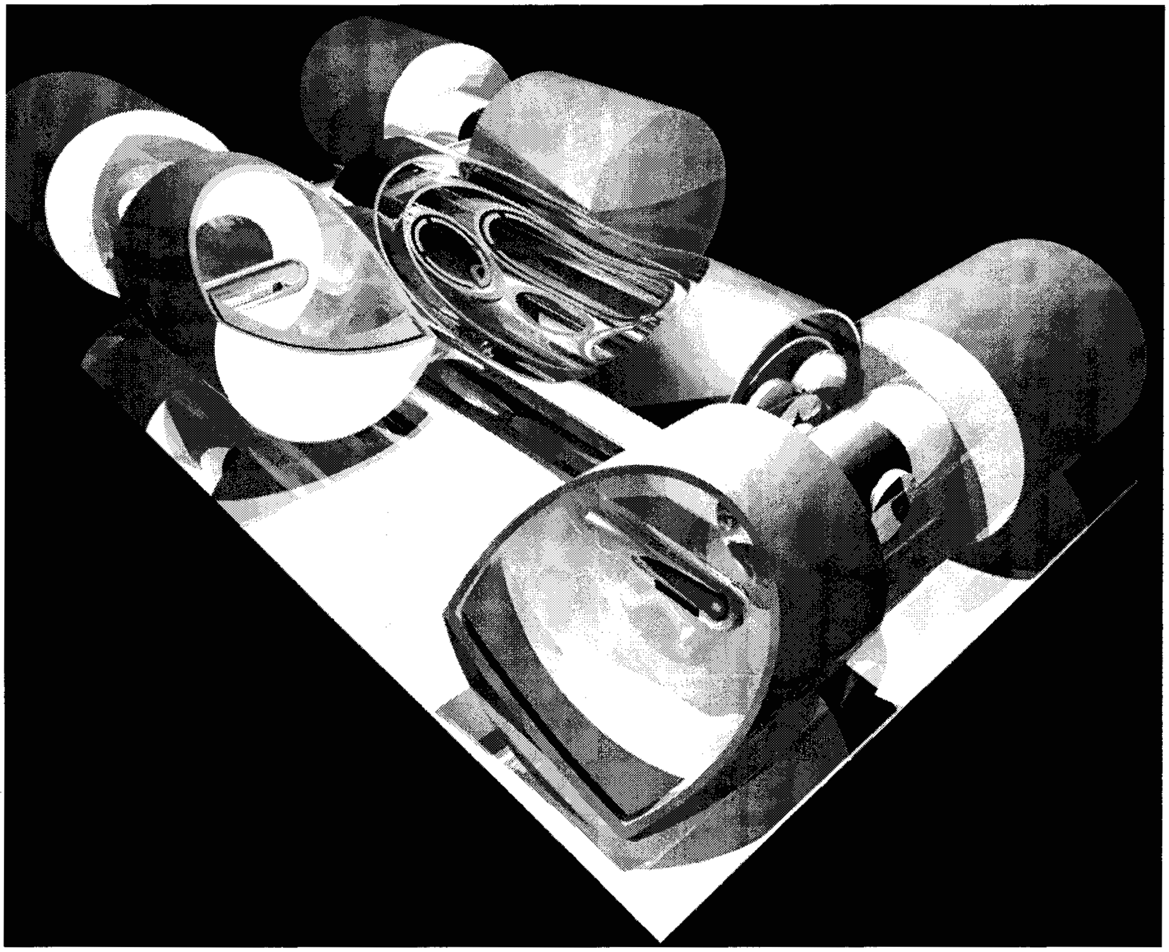
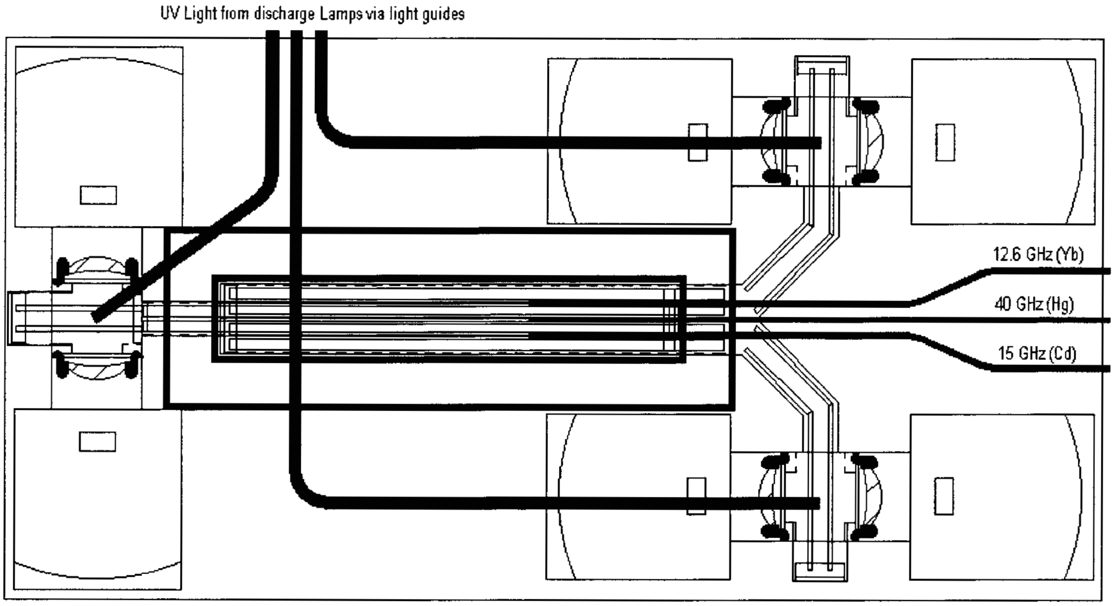


Figure 1-1. 3-D cutaway view of the SpaceTime atomic clock. This is the assembly view of the LITE mercury ion frequency standard. It is a modification of the linear ion trap frequency standard now being deployed in the Deep Space Network. The prototype of this clock has shown ultra-high frequency stability as good as the original unit in a much smaller package. The ion trap extends along the full length of the vacuum tube, from the optical flourescence region on the left end where lens and mirrors image section of the ion cloud onto ultraviolet photon counters, to the resonance region on the right where a uniform magnetic field is generated by a solenoid magnet wound on the vacuum tube wall. Four layers of magnet shielding surround this volume where the multiplied output of a voltage-controlled crystal oscillator resonates the clock transition in the mercury ion.

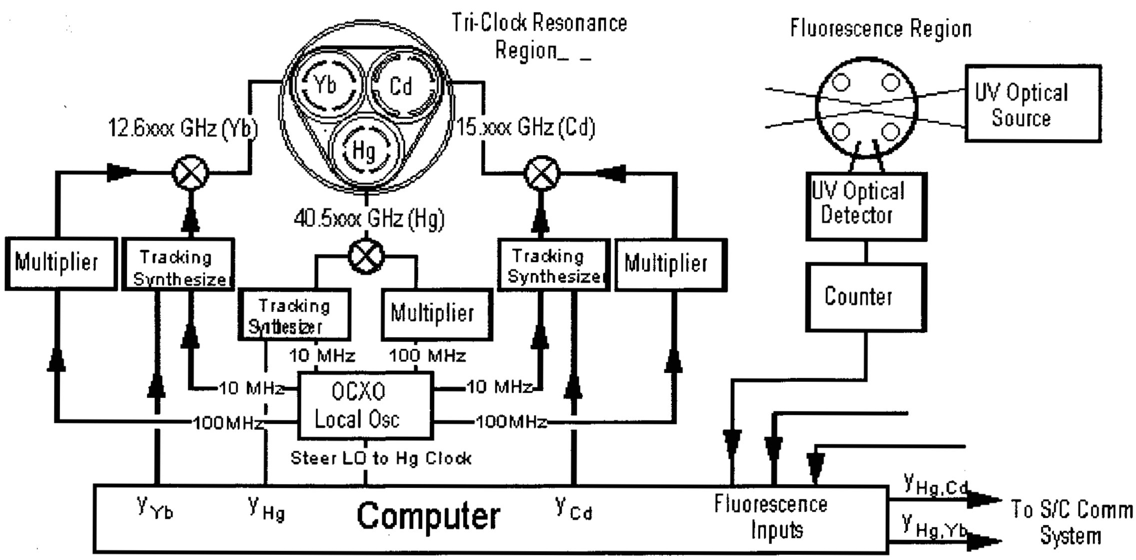
1.2.1.6 Modifications for Space Operation

For ultrastable operation the tri-clock standard will require temperature regulation to ±0.1 °C which must be realized within the power allotment (3 watts). The spacecraft payload interface platform where the clock will be mounted will be regulated to ±1 °C as part of the s/c thermal design. A factor of ten reduction of this base-plate temperature variation at the site of the physics package with time constant 100 seconds or longer will require thermal design unique to the space clock.

One of the most important thrusts in space qualifying the tri-clock lies in ruggedizing the physics package mechanical layout to withstand the acceleration loading of the 20-g launch. This will be



Figures 1-2 and 1-3. The tri-clock shown above (10×23×44 cm^3) has a common resonance region for simultaneously measuring the hyperfine clock transitions in Hg, Cd, and Yb ions. Two layers of magnetic shielding and a single solenoid magnet surround the three linear ion traps in the resonance region. Following optical state preparation is outside this region, ions are electrically 'shuttled' into the resonance trap. As shown below, a single local oscillator (LO) simultaneously interrogates each clock so that LO noise, normally the dominant noise source, is absent in the fractional difference frequencies $y_{Hg,Cd}$ and $y_{Hg,Yb}$.



done within the mass budget of 20 kg and with careful attention to mechanical resonance frequencies of the clock structure. In this effort we will use what has been learned in the GPS clock program where the same problems have been solved for cesium vacuum beam tubes. In fact, because the tri-clock physics package is similar in size and mechanical layout to the GPS cesium, this aspect of the space qualification should present no problems.

1.2.1.7 Ground-Based Environment Tests

The success of this mission will rely on extensive ground testing under conditions which replicate the spacecraft mission environment. Magnetic, thermal, vibration, and radiation tests will be carried out at the component level for some critical systems, such as radiation dose rate tests on the PMT uv light detectors. Before attachment to the spacecraft, the operational clock will undergo each of these four tests measuring frequency sensitivity to each controlled variable. Final ground-based testing will be carried out after the clock is integrated with the s/c and consists of calibrating clock frequency differences to s/c magnetometer readings. Following successful completion of these tests the s/c with the tri-clock inside will be depowered, shipped, and mounted on to the launch vehicle. Before launch, a final extensive set of clock system tests will ensure full operational status before launch. During launch, the clock will be depowered under sealed vacuum with no active pumping.

Clock environmental data, e.g., spacecraft magnetometer readings and base-plate temperature, along with clock-internal monitor voltages, e.g., lamp light output level, will be recorded continuously for the mission duration inside of ~2 AU. This data will also be compared to pre-launch test-chamber calibrations to remove any residual clock sensitivities to the space environment to the 10^{-16} level.

1.2.1.8 First Solar Flyby as a Pathfinder for Future Solar Probe Missions

Because this will be the first encounter with the solar environment as close as 4 solar radii, any in-situ measurements will provide a baseline for future missions. For example, spacecraft temperature will be measured in order to guarantee 10^{-16} resolution in the clock-difference frequencies. These data will be invaluable in characterizing s/c thermal performance, especially the carbon-carbon heat shield, the most important of the engineering developments for this and future solar missions.

The solar magnetic field along the trajectory will also be measured in order to aid in the frequency resolution of the clock inter-comparison. This scientific return will yield the first in-situ measurement of the solar magnetic environment.

The X-band communications link will also carry useful information about the intervening charged particle content.

1.2.1.9 Technology

This clock mission will develop and fly the most stable frequency standard ever flown, fully 100 times more stable than the present Global Positioning System (GPS) cesium and rubidium clocks. The space clock is a modified architecture of the LITS clocks now being deployed in the Deep Space Network stations worldwide. The principles of operation of the ion clocks have been demonstrated and represent the state of the art for ultrastable frequency standards. The strength of the space clock to be developed under this program is its ability to deliver the full stability of these ground-based units in a package comparable in mass and power to the GPS cesium clocks. This 100-fold improvement in stability in a small space qualified frequency standard would have a profound impact on the GPS navigation system and be a very attractive candidate for a GPS Block III system clock to lead Earth-based navigation into the 21st century.

The clock developed here would form the basis for an autonomous deep space navigation system where ultralight and ultrastable clocks would be onboard a s/c. Such deep space navigation systems, where a two-way Doppler link originates at the spacecraft, is sent to the ground stations and transponded back, would allow the spacecraft to navigate using only a fraction of the ground-station antenna time as is currently used. Such systems are now being planned by NASA for deep-space missions where ground-station antenna time allocation must dramatically shrink as small, inexpensive missions proliferate. With further development, this space LITE clock could evolve into a 5 kg, 10W deep-space clock delivering stability 10^{-15} over multiyear time intervals to dramatically change the technology of deep-space navigation.

Commercial development of the Hg+ LITE awaits the outcome of the packaging issues of a small clock and share many of the same challenges as would be addressed during the space clock development. The effort required to miniaturize the space LITE would be done jointly with Frequency and Time Systems, Inc., an industrial partner with extensive experience in GPS cesium space clock development. The industrial partner would gain familiarity with the LITE Hg+ technology and be well positioned to commercialize the clock having gained the confidence that can only be acquired through such a partnered development program with a working clock as the final product.

1.2.2. Mission

A prominent feature of the SpaceTime is its simplicity. Since our experiment consists of observing the influence of the solar gravitational potential on the rate of the tri-clock instrument, the mission requirements are simply "getting there," and the ability to transmit the clock rate data back to Earth. To satisfy the first requirement, i.e., reaching the proximity of the Sun, we will use an orbital trajectory to flyby Jupiter for a gravitational boost. After the Jupiter flyby, the solar trajectory is, for all practical purposes, determined. Since our science data quality depends on a gravitational potential gradient, the mission can yield useful data even if the trajectory deviates appreciably from the nominal design value. Details of the trajectory design and the requirement for possible trajectory correction maneuvers before Jupiter encounter are presented in Section 2-1. The radiation dose at Jupiter will be approximately 25 krad, so there will be no damage to either the spacecraft or the (powered down, at Jupiter) tri-clock during Jupiter encounter.

Our strategy for receiving data on Earth starts with a rudimentary recording and processing capability on spacecraft. This capability allows the data from the three atoms of the tri-clock instrument be compared every second, and then statistically processed to form a string of data once every 60 seconds, which will subsequently be recorded for transmission to Earth.

The spacecraft requirements include the ability to orient itself to thermally shield the Sun, and the capability to provide enough power to ensure proper operation of the instrument and the communications system. The issues of power and thermal shielding are the most important aspects of the spacecraft design and are discussed in detail in Section 2-3. We note, however, that if the Jupiter flyby leads to a trajectory which ultimately brings the spacecraft closer than four solar radii to the Sun, the science data will still be obtained and transmitted to Earth before the spacecraft experiences any damage.

Our strategy for data analysis, requires a knowledge of temperature and magnetic environment of the spacecraft bus housing the instrument. The design therefore includes several thermistors located at strategic spots in the spacecraft, and a three-axis magnetometer. The sensitivity of the thermistors are modest, at 0.1 °C; the magnetometer sensitivity at 1-mG resolution is adequate, though typically magnetometers have a resolution of a factor of ten better. The data transmitted to Earth include the thermal and magnetometer readings obtained once every minute.

2. MISSION IMPLEMENTATION

The SpaceTime mission enables first-of-a-kind science with a robust architecture while reducing mission risk through new applications of technology (see Tables 2-1 and 2-2).

Table 2-1. MISSION CHARACTERISTICS.

Feature	Benefit
Resilient launch and opportunity: nominal launch in Oct '02 with backup in Nov '03	Mission launch probability increased
Short deep space mission duration of 3.7 years	Can go single string on some subsystems with high probability of mission success
Proven Delta 7925 launch vehicle with standard Star 30c solid rocket motor kick stage	Reduces mission complexity and cost
Simple trajectory—Jupiter Gravity Assist (JGA, R_j= 8.68)	Reduced operations cost and complexity
Simple spacecraft using monopropellant propulsion system	Cost and risk reduction
Simple, low-rate X-band telecommunications via medium gain antenna (MGA)	Reduces operations and data processing costs
Beacon monitor, student supported operations	Low usage of oversubscribed DSN resources results in lowered mission cost
Light carbon-carbon weight heatshield	Allows 4 solar radii flyby, which provides adequate clock redshift signature while still leaving heat shield performance margin

Table 2-2. SpaceTime Overall Requirements.

Requirement	Units
Pointing knowledge	1°
Pointing accuracy	2°
Stabilization	3-axis
Propulsion System	Hydrazine monopropellant
Power System Mass	37 kg
Peak Power	108 W
Battery Type	DoD HED 300 Amp-Hr
Solar Array Type (low T/high T)	Silicon/GaInPh
Solar Array Area (low T/high T)	6.7/0.15 m^2
Array Mounting Type	Deployable, low T & high T
Downlink Data Rate	100 bps
Downlink Band	X-band
Transmitter Power	3W
Spacecraft Dry Mass	148 kg
Payload Mass (incl in s/c dry)	20 kg
Spacecraft Wet Mass	178 kg
Jupiter Encounter	8.68 R_j
Solar Encounter	4.0 solar radii periapsis
Expected Radiation Dose	50 krads (25K Jupiter fly-by, 25K rest of mission)

2.1 Mission Design

The goal of the SpaceTime mission design is to deliver the clock payload to the close vicinity of the Sun while minimizing complexity and risk.

SpaceTime launches from Cape Canaveral Air Station (CCAS) on a Delta 7925 launch vehicle (see Figure 2-1). This proven vehicle simplifies all aspects of the mission and allows us to reduce risk and cost, and return breakthrough science in the shortest time possible.

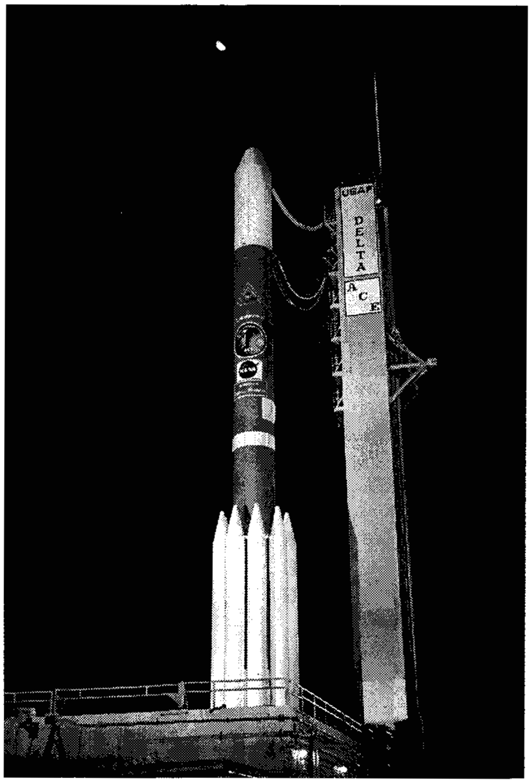


Figure 2-1. shows simple, reliable Delta 7925 in launch configuration. This launch system allows SpaceTime to meet the the cost, risk, and schedule of this MIDEX AO.

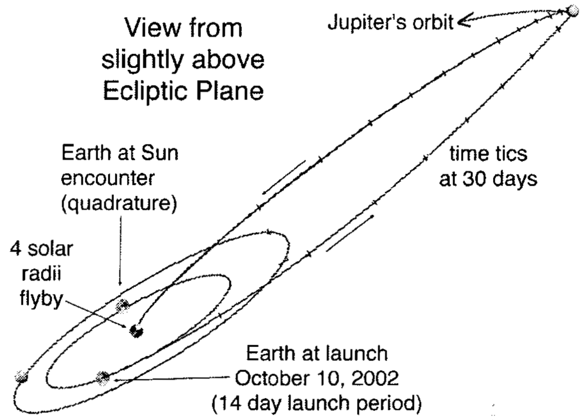


Figure 2-2. SpaceTime JGA shows simple trajectory that minimizes operations and returns new science as quickly as possible.

The launch period is 14 days long and opens on Oct 10, 2002. After the first two stages burn, the Star 48 third stage spins up to 60 rpm and injects the Star 30c/spacecraft towards Jupiter. As soon as possible after the Star 48 burnout, the Star 30c kick stage then ignites to supply the remainder of the required C_3 of 120 (km^2/s^2). A simple yo system despins the spacecraft to ± 0.5 rpm, where the attitude control system of the spacecraft can then 3-axis stabilize the spacecraft. At the conclusion of these solid motor burns the spacecraft is on its way towards Jupiter. After reaching Jupiter, a gravity assist is used to reduce the spacecraft energy, allowing it to fall toward the Sun and a 4 solar radii pass (Figure 2-2).

The SpaceTime mission plans for two trajectory correction maneuvers (TCMs). The first Trajectory Correction Maneuver (TCM) occurs at ~10 days after launch. This TCM could be as much as ~170 m/s (3σ) and is used to correct launch injection errors. Then, Doppler tracking is collected every two weeks to one month. Ranging is not needed, which simplifies the spacecraft telecommunications subsystem. The final TCM will be approximately one year after launch. This small clean-up targets the spacecraft to the Jupiter flyby aimpoint at 8.68 R_J.

Since SpaceTime differential redshift measurements allow a 10% error in the solar flyby radius, we target to 4.2 $R_s \pm 0.2\ R_s$. This allows tolerable large navigation uncertainties at the Jupiter flyby. Also, 1°–2° errors off quadrature (90° inclination solar flyby with Earth perpendicular to orbit plane at solar flyby) are acceptable since the medium gain antenna has an ~5° beam width.

After the Jupiter encounter, the spacecraft falls rapidly toward the Sun (see Table 2-3). No further TCMs are required, although slight corrections are possible if required. The greatly simplified navigation and tracking scheme allows for a Doppler track once every 1–3 months. This simplifies mission operations and reduces cost at no increase to mission risk.

Table 2-3. Typical Trajectory Chronology.

Event	Date	Time
Launch	16-Oct-2002	---
1.5 AU crossing	30-Dec-2002	15:0612
2.0 AU crossing	15-Feb-2003	08:13:45
3.0 AU crossing	24-May-2003	00:25:08
Jupiter flyby (8.68 $R_{Jupiter}$)	15-Mar-2004	16:26:39
Max Earth to s/c distance (6.3 AU)	Aug 2004	---
3.0 AU crossing	09-Dec-2005	10:46:01
2.0 AU crossing	08-Mar-2006	20:38:35
1.5 AU crossing	11-Apr-2006	03:23:57
1.0 AU crossing	07-May-2006	13:41:22
0.7 AU crossing	20-May-2006	02:26:08
0.3 AU crossing	01-Jun-2006	11:58:38
0.2 AU crossing	03-Jun-2006	16:29:57
0.1 AU crossing	05-Jun-2006	09:51:19
0.05 AU crossing	06-Jun-2006	01:34:34
10 Rs crossing	06-Jun-2006	02:32:35
9 Rs crossing	06-Jun-2006	03:49:09
8 Rs crossing	06-Jun-2006	05:04:51
7 Rs crossing	06-Jun-2006	06:20:36
6 Rs crossing	06-Jun-2006	07:38:22
5 Rs crossing	06-Jun-2006	09:03:51
Perihelion (4 Rs)	06-Jun-2006	11:47:14
5 Rs crossing	06-Jun-2006	14:30:37
6 Rs crossing	06-Jun-2006	15:56:07
7 Rs crossing	06-Jun-2006	17:13:52
8 Rs crossing	06-Jun-2006	18:29:37
9 Rs crossing	06-Jun-2006	19:45:20
10 Rs crossing	06-Jun-2006	21:01:53

2.2 Instrument Accommodation

By design, SpaceTime has few requirements placed upon the mission architecture by the atomic clock payload.

The only requirements placed upon the SpaceTime mission by the clock payload are:

- Adequate warm-up and calibration time before the crucial solar gravity field measurements
- Protection from adverse temperatures in the clock electronics during the cruise phase of the mission (±40 °C)
- Maintain the payload interface to the spacecraft when the clock is operating to ±1° C
- Minimize electric and magnetic fields produced by the spacecraft
- Return 60 bits/sec of differential frequency science data during the solar passage

These requirements and a description of how they are accommodated are summarized in Table 2-4.

Table 2-4. Instrument accommodation summary shows few requirements. This allows simplification and consequent risk and cost reduction.

Clock Requirement	Accommodated by:
30 days of warm-up and calibration	Adequate power before Sun flyby
Non-operating range of ±40 °C	Heater power and simple, robust thermal design
Maintain payload interface to ±1 °C	Design of thermal subsystem
Minimize electric and magnetic fields of spacecraft	Proper grounding and material choice for spacecraft
60 bits/sec downlink science data	Medium gain antenna telecomm system at X band
Payload data	Serial interface to spacecraft computer

2.3 Spacecraft

The SpaceTime spacecraft design is simple and reliable. This design leads to reduced cost and risk in mission operations and allows the science data return at its critical point near the Sun.

The SpaceTime spacecraft utilizes extensive heritage from past spacecraft, including Mars Pathfinder, Cassini, DS-1, Lewis, MSP '98, and Defense Systems programs. Comprehensive analysis has produced the best small spacecraft design for the SpaceTime mission within the budget constraints. The primary purpose of the spacecraft is to take the SpaceTime clocks as close as possible to the Sun and to allow the acquisition and transmission of data from the clocks as the s/c nears the Sun. These analyses confirm that the proposed design with some functional redundancy meets the challenging mission requirements.

Figures 2-3 and 2-4 show a spacecraft close-up and a near-solar pass configuration. The spacecraft integrates a blowdown monopropellant propulsion system and the necessary spacecraft electronics to support the mission. The spacecraft electronics and the clock are all within a single thermal enclosure. The propellant tank and thrusters are colocated at the aft end of the spacecraft. The large, conical solar shield protects the entire spacecraft during the 4 solar radii encounter (see Section 2.4 for further heat-shield details).

Figure 2-5 is the flight system block diagram that shows subsystem components and defines all top-level power and data interfaces. The spacecraft is spin stabilized following launch vehicle separation and the first TCM to minimize early operation time on the single string components. The normal cruise configuration for the spacecraft outside the Earth's orbit will be in a slow spin with the low temperature solar array fully extended. On its final trajectory towards the Sun, the spacecraft is 3-axis stabilized inside of Earth's orbit for the close solar flyby. The conical shield is nadir pointed at the Sun with a 2° pointing accuracy.

Table 2-5 displays the detailed SpaceTime Spacecraft Mass, Power and Heritage List. Selective self shielding utilizing system components as much as possible for the C&DH and Power System electronics are required to mitigate the total radiation dose. The SpaceTime spacecraft current best estimate dry mass is 114 kg, including the 20-kg science payload, with contingencies of 30% in both mass and power. The fuel load is about 30 kg for a launch mass of around 185 kg, the maximum allowable mass that can be launched on a Delta 7925 with Star 30c kick stage to a C3 of 120 km^2/s^2.

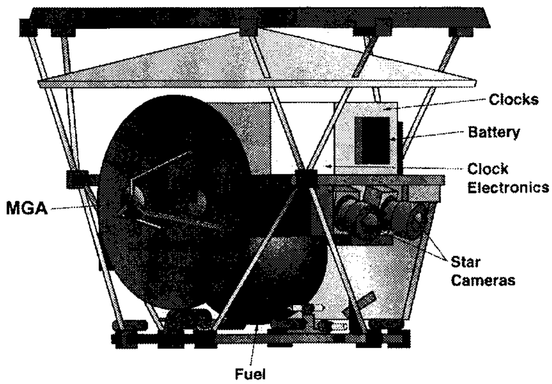


Figure 2-3. Close-up view of spacecraft.

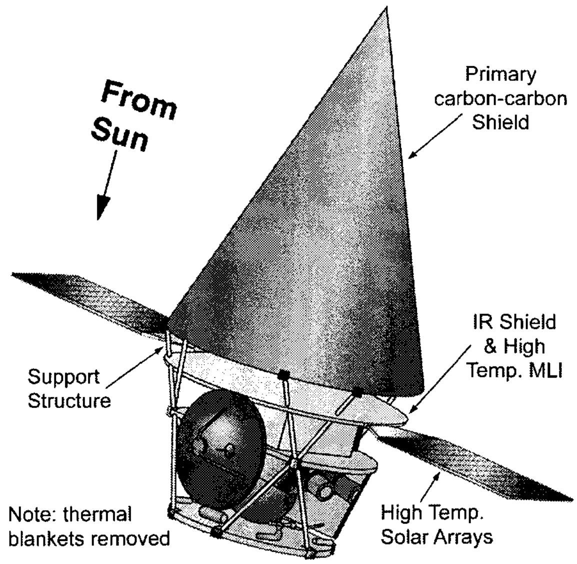


Figure 2-4. 3-D view of high temperature array spacecraft.

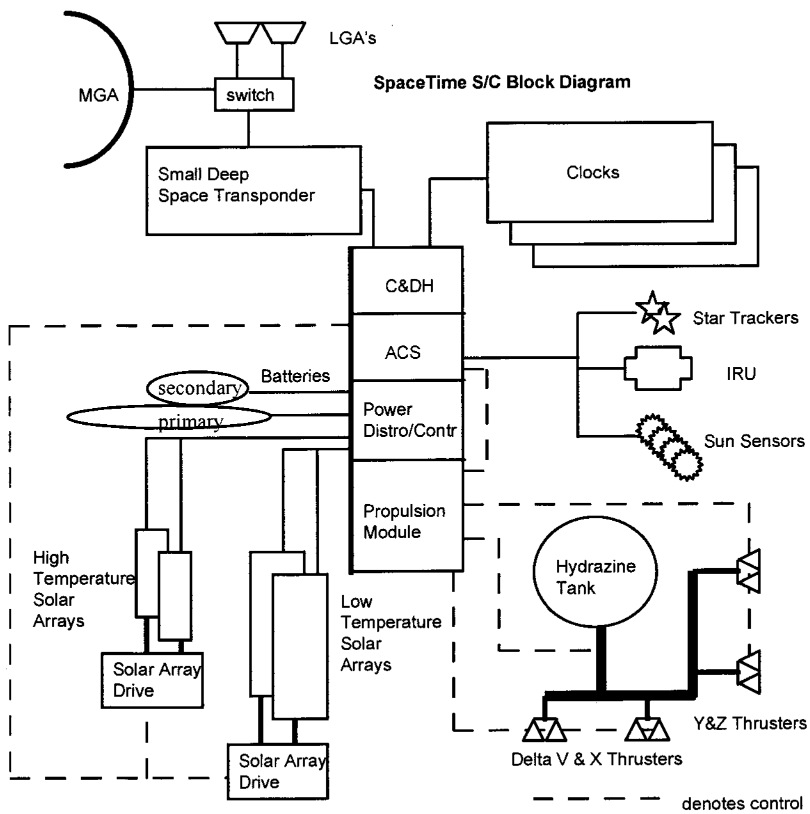


Figure 2-5. Flight system block diagram.

The attitude control subsystem (ACS) subsystem provides attitude knowledge and control, and solar array articulation and control. The Medium Gain Antenna (MGA) is body-mounted, perpendicular to the spacecraft centerline. Twin star trackers and an Inertial Measurement Unit (IMU) provide primary attitude determination, with four analog Sun sensors serving in a functional redundancy role as backup pointing devices. During the Jovian gravity assist, the ACS system will operate in an all-stellar mode so that power can be conserved by turning off the IMU.

The propulsion system uses a blowdown system with monopropellant hydrazine and GN2 pressurant. The subsystem design employs four pairs of 0.9 N thrusters for all ACS and TCM maneuvers.

The Command and Data Handling (C&DH) subsystem provides instrument science data storage, spacecraft commanding, interfaces to the X-Band transponder, ACS sensors and actuators, and spacecraft timing. The C&DH uses a proven RAD-6000 processor, I/O, payload/attitude control interface (PACI), uplink/downlink cards, 25 MBytes of DRAM storage and 3 Mbytes of EEPROM. The flight software load is stored in EEPROM in compressed format. Because of the real-time data requirement, very little data storage is required for the C&DH. The RAD 6000 processor flew in a similar single string configuration on the Mars Pathfinder mission.

The communications subsystem supports X-band command uplink and data downlink using the Small Deep Space Transponder, a 3-watt Solid State Power Amplifier (SSPA), one 0.6-meter diameter MGA, and dual low-gain patch antenna (LGA). The MGA provides 100 bps data transmission near Jupiter and 100 bps during the solar flyby. The LGAs provide emergency command capability. The SDST flew successfully on DS-1 and on the MSP '98 orbiter. See Table 2-6 for further telecomm subsystem details.

The power subsystem provides power collection, storage, control, and distribution. The Electrical Power Subsystem (EPS) is comprised of the solar array, battery, charge control unit (CCU), power distribution and drive unit (PDDU), and pyro initiator unit (PIU). Because of the profile of this mission, the SpaceTime spacecraft carries two sets of featherable solar arrays, a small secondary battery and a high-energy primary battery for the critical final phase of the mission when the solar arrays can no longer operate. The low temperature solar panels are populated with 6.7 m^2 of silicon solar cells, providing 1500 W average power near Earth. The low temperature solar arrays are sized for stay-alive power near Jupiter. Temperature and power control near Earth for the spacecraft will be accomplished through the cosine loss of the low temperature featherable array. The low temperature array is jettisoned after crossing Mercury's orbit at 0.3 AU. The high temperature portion of the solar array consists of 0.15 m^2 of Gallium Indium Phosphide cells at 6% efficiency and provides power after crossing 0.3 AU. These arrays will provide up to 200W of power as close as 0.1 to 0.15 AU when feathered to a 70-degree angle of incidence. This will allow the SpaceTime clocks to be calibrated prior to the final phase of the experiment. The high temperature arrays will also be jettisoned to prevent an inbalance in control forces near the Sun. The High-Energy Density (HED) primary battery, a 300 A-hour Lithium Carbon Fluoride (LiCF) package with a long shelf life, provides the 3730 W-hr necessary during the 33-hour solar flyby after the solar arrays are jettisoned at about 0.13 AU.

The structural configuration includes the integrated propulsion and electronics subsystems, solar array substrates, and the MGA attachment hardware. The truss design provides a lightweight structure with nearly unrestricted access. The structure supports loading from the fuel tank, electronics and payload, Solar Array (SA) assembly, MGA assembly, and thruster assemblies. Launch loads are transmitted to the LV via the V-band adapter ring.

The mechanisms consist of 4 single-axis gimbal assemblies for the four wings of the SA; spring-driven, single-axis deployment hinges between the SA panels deployed by conventional retention; and release devices which protect the SA for launch loads.

Table 2-5. Flight System Mass and Power.

Subsystem	Qty	Each kg	CBE kg	Power	Flt Heritage/Descrip.
AACS					
Star Tracker	2	2.68	5.36	5.9	Lewis
IRU	1	0.70	0.70	12.5	Clementine
Sun Sensors	4	0.14	0.56	0.3	Iridium (EDO Barnes)
Solar Array/ Prop Valve Elec.	1	0.80	0.80	1.3	Lewis
Total			7.42	20.0	
C&DH: RAD 6000	1	2.00	2.00	6.0	Pathfinder
Telecom					
X-Band SSPA, (3W)	1	0.50	0.50	5.0	Pathfinder
X-Band Transponder (SDST)	1	3.10	3.10	6.0	DS-1
MGA/LGA Select Switch	2	0.37	0.74		Pathfinder
X-Band Diplexers/WVGD/splitter	1	1.55	1.55		Pathfinder
LGA Wave guide	1	1.00	1.00		DS-1
LGA (X-Band)	2	0.04	0.08		MGS
MGA Antenna	1	0.86	0.86		
Misc (coax cable, connectors, filters)	1	1.00	1.00		Numerous
Total			8.83	11.0	
Power					
Cruise Solar Array Assembly	2	7.50	15.00		Silicon: Numerous
High Temp Solar Array Panels	2	0.88	1.76		GaInPh
Power Distribution & Control Unit	1	1.91	1.91	3.0	Lewis
Connectors, Internal Cables	-	0.20	0.20	3.0	Lewis
Primary Battery	-	12.00	12.00		DoD HED 300 Amp-Hr
Secondary Battery	-	4.70	4.70		Numerous
Pyro Switching Unit	2	0.50	1.00		Lewis
Deploy/Sep Mech. Low Temp	2	2.50	5.00	4.0	
Deploy/Sep Mech. High Temp	2	1.25	2.50	2.0	
Total			36.57	12.0	
Thermal					
Shield - Primary: 1m or less	1	5.00	5.00		Solar Probe
HGA/Shield Support Struts	4	0.20	0.80		C-C tubing
Shields - Secondary IR #1, #2 & #3	1	4.90	4.90		C-C sandwich
Support, Stiffeners, Fittings,etc.	-	1.00	1.00		C-C, Ti & Graphite-Epoxy
MLI/Louver	-	2.00	2.00		Cassini
Temp. Sensors, Heaters, Misc.	-	1.00	1.00	5.0	perihelion
Total			14.70	5.0	
Structure					Numerous
Total			9.57		
Propulsion					Cassini
Total			9.57	1.0	
Cabling: Total		5.50	5.50		(~6% dry mass exc heat shield)
Payload: Tri-clock		20.00	20.00	28.0	
Total (CBE)			**114.16**	**83.00**	**Peak Power in Watts**
System Contingency (30% of total CBE Masses)			34.25	24.90	Contingency
Dry Spacecraft with Contingency			**148.41**	**107.90**	**Total**
Propellant (N2H4)			30.39		15-kg ACS; 15-kg DV
GN Pressurant	0.2		0.17		
Wet Spacecraft			**178.97**		
Upper Stage Adapter(using Star 30C U/S)~3% of Wet Mass			5.37		
Total Launch Mass w/System Contingency of 30%			**184.34**		
Delta II 7925/STAR 30C - Oct, 2002 (GSFC OLS performance quote)			**185.00**		
Launch Margin w/ System Contingency			**0.66**		

Table 2-6. Required Telecommunications Parameters and Definitions.

Parameter	Units	Description
Maximum S/C Distance	Km	5.4 AU after Jupiter gravity assist
Jupiter Gravity Assist Flyby Distance	Km	5.2 AU (8.68 Rj flyby distance)
Sun Encounter Distance	Km	1 AU S/C ñ Earth distance (4 solar radii flyby, quadrature with ~5_ MGA pointing
Uplink Transmitter Power	Watts	34 meter HEF
Uplink Frequency Band	GHz	7.170 GHz.
Uplink Transmitting Antenna	dBi	34 meter HEF
S/C Receiving Antenna Gains	dBi	Medium gain ~ 20, LGA gain ~ +6
Telecommand Data Rate	b/s	256
Telecommand Bit-Error-Rate	-	1 X 10^{-6}
S/C Receiver Bandwidth	Hz	10 Hz (SDST transponder)
Turnaround Ranging	Yes/No	No
SC Transmitting Power	Watts	3 Watts RF
Downlink Modulation Format	Name	PCM/PSK
Downlink Frequency Band	GHz	8.450 GHz.
S/C Transmitting Antenna	dBi	Medium gain ~ 20
Downlink Receiving Antenna	dBi	34 HEF for cruise 70 meter DSN
Telemetry Data Rate	b/s	100 bps
Error Detecting-Correcting	Name	7, 1/2 Reed-Solomon/Viterbi concatenated coding scheme
Telemetry Bit-Error-Rate	-	1 X 10^{-5}

The SpaceTime thermal control system (see Section 2.4) features a 1.3-meter diameter Solar Thermal Shield (STS) to protect the spacecraft during solar swingby. The lightweight STS shadows the structure and equipment during solar swingby so the instruments and electronics can view cold deep space. Electrical strip heaters are used for heating of selected components such as thruster valves.

2.4 Heat Shield

The thermal control system of the SpaceTime spacecraft will utilize a thermal shield system that will isolate its flight elements from the solar flux of a 4 Rs (from the solar center) close passage. This shield system is a technology development and consists of a solar blocking element (primary shield), IR shields, High Temperature MLI, and support structure. The system is a simpler version of the Solar Probe shield system, and will use their test and design parameters to minimize risk. The first element of this shield will operate at about 2000 K during close solar approach, but the shield system will provide thermal isolation that will allow the spacecraft elements behind it to operate at nominal temperatures (0 to +40 °C).

The spacecraft thermal control system components behind the shield system utilizes flight-proven elements such as thermal conduction isolation, thermal surfaces (paints, films), MLI, louvers, and electric heaters/thermostats. This design will control spacecraft element temperatures for all flight regimes from launch to Jupiter flyby, and close solar approach.

2.5 Communications Approach

SpaceTime communications utilize an appropriate beacon monitor mode and a simple X-band RF subsystem to reduce the amount of required tracking while maintaining a low level of risk during operations.

After launch, a period of two weeks of continuous tracking on the 34-meter DSN subnet will be utilized for spacecraft checkout, injection, clean-up, TCM preparation, and clock calibration (see Table 2-7). Over 90% of the mission is operated in beacon mode. This "near-hibernation" mode allows the spacecraft to contact flight controllers only when necessary. This allows minimized use of single- string components and reduced DSN costs. Beacon mode is being demonstrated on the DS-1 mission.

Table 2-7. DSN tracking characteristics show greatly reduced requirements (and cost) while maintaining a robust, reliable mission.

Mission Phase	DSN subnet	Tracking freq, hrs/track
Launch (2 wks)	34 meter	continuous
Cruise	34 meter*	1/mon, 1 hr
Maneuvers	34 meter	2/mission, 4 hrs
Nav data	34 meter	1/mon avg, 8 hrs
Sun Encounter	70 meter	Continuous, 2 wks

*Will study using smaller stations for beacon monitor tracks such as the 18 meter dishes outside Boulder, Colorado

2.6 Mission Operations Plan

SpaceTime uses proven mission operations personnel, procedures, and facilities at CU and JPL to reduce mission risk and provide science data in near-time after receipt.

The mission operations team for SpaceTime will be distributed to take advantage of expertise and to reduce operations costs. Mission Control will be accomplished at the University of Colorado's Space Grant Program where students and staff will be responsible for generating operational sequences, for controlling operations, monitoring performance, maintaining the ground data system, and interacting with the science and engineering teams. Engineering experts from JPL will support the Mission Control team by diagnosing any spacecraft problems, calibrating long-term performance, and maintaining spacecraft performance.

The operations team at JPL will include the science team, the clock payload lead, and spacecraft experts. This team will be responsible for the long-term performance of the clock payload, for spacecraft navigation, and for the performance of the telecommunications and thermal subsystems.

Operations costs will be minimized by taking advantage of the reduced levels of activity expected during the cruise phase—which accounts for approximately 90% of the mission. During the low activity cruise phase, DSN tracking will be scheduled only once each month for support of spacecraft performance checks, maintenance and navigation tracking (beacon monitor mode). Between these monthly checks, the health and status of the spacecraft will be monitored autonomously on-board. If the spacecraft autonomy determines that help is needed, a request beacon will be sent to the ground network and the ground operations team will be notified immediately.

Since the clock Equivalence Principle experiment produces analyzable redshift frequency difference data only near the Sun, this data will be recorded at two DSN complexes, when possible, in the 10 hours before perihelion. On-board battery power limits mean that data cannot be played back after the solar encounter. So, the data will be downlinked at 100 bps (total rate, 60 bps science) and recorded at two separate DSN locations. Initial analysis shows that the Goldstone and Canberra 70-meter stations are suited for this purpose.

2.7 Conclusion

The SpaceTime science objectives and mission just described are the culmination of work for a NASA MIDEX 98 proposal from JPL. While SpaceTime was not selected to move forward to project formulation, the concept of using space atomic clocks flying close to the sun for a unique test of Einstein's Equivalence Principle is promising. The mission was deemed too risky because current atomic clocks and carbon-carbon heat shield technologies are not mature enough for the cost and schedule constrained MIDEX program yet. The proposal team expects that these technologies will mature over time and that a derivative of SpaceTime will be submitted for the MIDEX 2000 mission competition.

2.8 Acknowledgments

I would like to thank the many talented individuals of the SpaceTime proposal team (John Anderson, John Armstrong, Freda Cheung, Elaine Hansen, Rob Maddock, Lute Maleki, Bob Miyake, Bill Moore, Ken Nordvedt, John Prestage, George Sprague, Ellie Trevarthen, Bob Vessot, Jay Wyatt and numerous others). This paper, the proposal, and the exciting prospect of new discoveries would not have been possible without them. Finally, I would like to thank my production assistant Mary Young for helping me create this paper.

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